

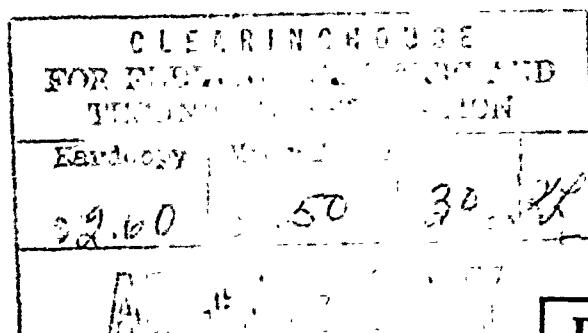
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DESCRIPTION OF THE 120-INCH HYPERSONIC SHOCK TUNNEL AND ANTICIPATED PERFORMANCE

PHILIPPE O. BOUCHARD, CAPTAIN, USAF
HAROLD F. CHAMBERS, JR.

TECHNICAL REPORT AFAPL-TR-65-129



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FOREWORD

This report was prepared by the Ramjet Components Branch, Ramjet Engine Division, Air Force Aero Propulsion Laboratory, Research and Technology Division, Wright-Patterson Air Force Base, Ohio. The work was conducted under Task 301201 of Project 3012, with Mr. H. Clyde Long as project engineer. The shock tunnel was designed and constructed by AVCO Corporation, Wilmington, Massachusetts, under Contract AF33(615)-66.

This report was submitted by the authors on 22 December 1965.

This technical report has been reviewed and is approved.

Robert E. Roy

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Laboratory

ABSTRACT

A 120-inch hypersonic shock tunnel has been installed in the Air Force Aero Propulsion Laboratory to provide a capability for testing advanced ramjet engine components and large-scale propulsion systems. The shock tunnel utilizes driver gases that are unheated; presently, either hydrogen or helium, pressurized to 30,000 psia, is used as the driver gas. The "tailored" interface technique is used to allow for maximum testing time. The facility in its present configuration yields test Mach numbers in the range 18 to 35. Modifications are being made to extend this range to approximately Mach 8 to 10.

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LIST OF SYMBOLS

SYMBOL	DESCRIPTION	UNITS
a	speed of sound	ft/sec.
C_p	specific heat at constant pressure	BTU/lb °R
C_v	specific heat at constant volume	BTU/lb °R
h	static enthalpy	BTU/lb
M	Mach number	
M_s	shock Mach number	
p	static pressure (absolute)	lb/ft ²
q	dynamic pressure	lb/ft ²
\dot{q}	heat transfer rate	BTU/ft ²
s	entropy	BTU/lb °R
t	time	sec.
T	static temperature	°R
v	velocity	ft/sec.
γ	ratio of specific heats (C_p / C_v)	
ρ	density	lb/ft ³

SUBSCRIPTS

o	stagnation conditions
$1, 2, 3$	static conditions at designated points
∞	free stream conditions
$*$	sonic conditions

SECTION I

INTRODUCTION

The Ramjet Engine Division of the Air Force Aero Propulsion Laboratory is conducting an intensive analytical and experimental effort to develop advanced ramjet engines for future hypersonic vehicles. Among these engines is the supersonic combustion ramjet engine, known as the SCRAMJET. One of the facilities capable of simulating flight conditions for hypersonic research is the shock tunnel. A 120-inch hypersonic shock tunnel, designed and constructed for the Air Force by the AVCO Corporation, was procured to further hypersonic aerodynamic investigations and supersonic combustion studies. Tests will be performed on both engine components and large-scale integrated propulsion systems.

This report provides a complete description of the shock tunnel components and instrumentation equipment, a brief review of the shock tunnel operating principles, and the expected performance capabilities of the facility. Although calibration of the facility is not yet complete, this information is furnished to aid interested agencies in planning test programs to be conducted in the facility.

SECTION II

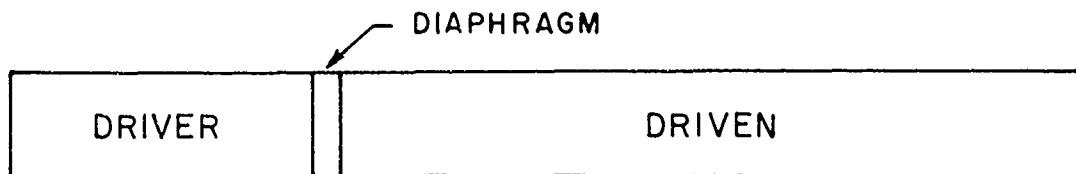
OPERATING PRINCIPLES

The shock tunnel can be considered as a very short-duration wind tunnel. It consists of a shock tube to process the test gas to a high stagnation pressure and enthalpy and a nozzle to expand the test gas to desired hypersonic conditions.

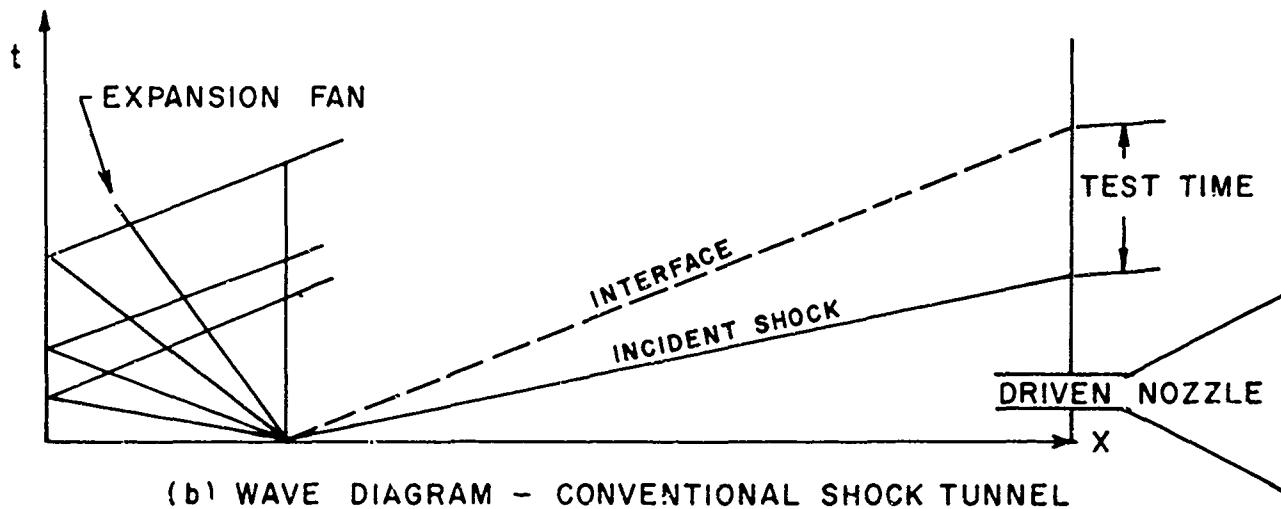
The shock tube is separated into regions of high and low pressure gases (the driver and driven sections, respectively) by a gas-tight diaphragm, as shown in Figure 1a. When the diaphragm ruptures, the high-pressure driver gas expands into the driven section, thereby causing a shock wave to propagate through the low-pressure gas. Between the fast-moving shock wave and the slower-moving gas interface there exists a region of steady flow which is at elevated pressure and enthalpy. The downstream end of the driven tube is terminated by a convergent-divergent nozzle, and the ratio of the throat area of this nozzle to the cross sectional area of the driven tube is small. This ratio is kept small so that the primary shock wave is reflected upstream from the throat section and passed through the already shock-treated gas, which results in a further increase in pressure and enthalpy. The gas behind the reflected shock is at rest relative to the shock tube. This reservoir of processed air then is expanded through the nozzle to the desired test conditions.

By controlling the initial pressure and temperature in the driver and driven sections, the reflected shock can be made to pass through the gas interface, that is, no gas dynamic waves result from this interface-shock interaction that can subsequently disturb the steady test-air supply conditions. This method is called the "tailored-interface" operation (References 1 and 2) because the states of the gases on both sides of the interface must be precisely matched. Tailored-interface operation increases testing time by almost an order of magnitude over nontailored operation. The useful test time ends when either the head of the expansion reflected from the upstream end of the driver or the interface arrives at the nozzle throat section, as shown in Figure 1b and c.

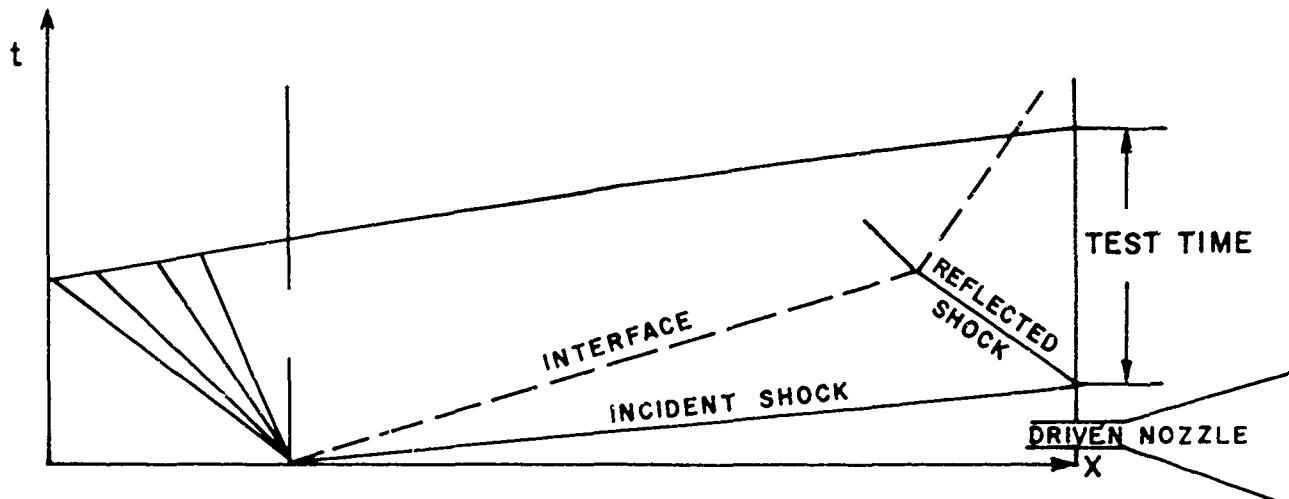
Although the tailored-interface technique is the most practical method of attaining the longer test times without causing excessive shock-wave attenuation, the technique has an upper bound in producing high stagnation enthalpy. Tailoring, which implies set initial driver/driven gas conditions, also results in a single "tailored" shock Mach number (M_S) for given driver/driven gases. With a helium/air combination for the driver/driven gases, the tailored shock Mach number is approximately 3.7. If a higher stagnation enthalpy is required than that resulting from an $M_S = 3.7$, then the speed of sound in the driver gas must be increased. This is normally accomplished by heating the driver gas or by using a lighter gas such as hydrogen, which has a "tailored" $M_S = 6$ (Reference 3).



(a) SCHEMATIC - SHOCK TUBE



(b) WAVE DIAGRAM - CONVENTIONAL SHOCK TUNNEL



(c) WAVE DIAGRAM - REFLECTED "TAILORED" SHOCK TUNNEL

Figure 1. Shock Tunnel Wave Diagram

SECTION III

SHOCK TUNNEL FACILITY

The general arrangement of the facility's major components is shown in Figures 2 through 5, and are here described in detail.

1. DRIVER SECTION

The driver section, shown in Figure 2, consists of two fifteen-foot long, thick-walled cylinders, capable of containing gas at pressures up to 30,000 psia for extended periods. Integral forged flanges incorporating rubber O-rings and metallic antiextrusion devices are used at the ends of each tube to form joints of high mechanical integrity and a positive seal.

Hydrogen can be used at high pressure in this driver, but the use of hydrogen causes embrittlement of the inner bore, which would result in driver tube failure. To prevent this type failure, a 4-inch inner diameter liner made of stainless steel, which is much less susceptible to hydrogen embrittlement, is inserted in the tube. A blind flange at the beginning of the tube accommodates a piping junction block which includes inlets for high pressure gas, nitrogen purging, vacuum pumping, and a safety relief.

Hydrogen, helium, or nitrogen can be supplied to the driver tube at 2400 psia, regulated to 1100 psia, and successively compressed in three stages: first stage, 6000 psia; second stage, 15,000 psia; and third stage to 30,000 psia. This pressurization process is accomplished by a three-stage Corblin compressor system.

2. DOUBLE RUPTURE DIAPHRAGM SYSTEM AND TRANSITION SECTION

The double-rupture diaphragm system contains two stainless-steel rupture diaphragms and the associated seals, as shown in Figure 3. A double diaphragm system is used because it provides a means of controlling the time of actuation of the shock-tunnel run and nearly halves the pressure required to burst each diaphragm. The diaphragms are designed to burst at pressures varying from 2000 to 18,000 psia in 2000 psia increments. The clamping force required to produce a positive seal upon the assembly is achieved by utilizing a coupling nut with an electric motor run-up and a secondary system for tightening. This system combines relative speed of operation with simplicity of design.

The original design of the facility employed a heated driver tube of 4-inch internal diameter and a driven tube of 6-inch internal diameter. Aft of the double diaphragm assembly, a conical transition section was incorporated to provide gradual enlargement from the driver to the driven tubes. This design was not changed in the conversion to high-pressure, unheated driver operation.

3. DRIVEN SECTION

The driven section consists of four 15-foot-long thick-walled cylinders having a 6-inch inner diameter as shown in Figure 4. These cylinders are capable of withstanding a steady internal pressure of 30,000 psia. Integral forged flanges incorporating rubber O-rings are used at the ends of each tube. No protective

liner is used in this section because it is exposed to high-pressure hydrogen for only a short period of time. The last portion of the driven section is a 6-foot-long thick-walled cylinder capable of withstanding 45,000 psia internal pressure. This is where the reservoir of test gas created by the shock is located. This tube is lined with stainless steel to protect the bore from embrittlement and is easily replaced in case of damage from heat.

A low pressure gas system supplies the test gas to the driven section. The gas is stored at 2400 psia, and this pressure is reduced to 500 psia or less at the gas manifold. Actual pressure maintained in the driven section depends on the particular test being run.

4. NOZZLE

The nozzle as shown in Figure 5, expands the reservoir gas to hypersonic Mach numbers in the test section. The 15-degree half angle conical nozzle has an exit diameter of 10 feet and three interchangeable throat inserts having throat diameters of 0.25, 0.50 and 1.00 inch. The nozzle is constructed in three sections: the first is 3 feet long with an exit diameter of 20 inches and is fabricated of stainless steel; the remaining two sections are fabricated of fiber glass and have lengths of 6 and 9 feet with exit diameters of 60 and 120 inches, respectively. The driven tube is coupled to the stainless-steel section of the test nozzle by a coupling nut; an electric motor is used for run-up and a secondary system for tightening. Normally, one or two thicknesses of 0.001-inch-thick Mylar make up the thin diaphragm at the entrance of the test nozzle.

5. TEST SECTION AND EXPANSION TANK

The 10-foot diameter nozzle exit section is, in effect, the upstream end of the 35-foot long expansion tank, which is long enough to prevent reflected disturbances from reaching the test region before the test time provided by the shock tube is terminated. Three 12-inch-diameter Schlieren observation ports are located just aft of the nozzle exit; an entrance hatch door is located further downstream of the windows.

6. MODEL SUPPORT SYSTEM

The model support and angle-of-attack adjustment system is comprised of a vertical sector, machined to a radius whose center is at the test station and which is placed aft of the test station by the amount of that radius, along which rides the model support sting. Model positioning is manual, and is accomplished within the tank. Maximum angle of attack with a straight sting is 45 degrees above or below the center line of the shock tunnel. That portion of the facility floor upon which the model support stands has been isolated with a cork material to minimize the transfer of vibration to the model being tested. This in turn is isolated from the tank by a bellows system.

7. VACUUM SYSTEM

The vacuum system consists of two Kinney, single-stage, cam and piston type vacuum pumps and one Consolidated Vacuum Corporation oil diffusion pump. This vacuum system is used to evacuate both the test section and the shock tube. This vacuum system is capable of evacuating the expansion tank down to 1 micron Hg in 35 minutes.

8. SUPPORTING STRUCTURE

The entire shock tunnel, with the exception of the expansion tank, is mounted on wheeled carriages. A chain-drive separation system provides motion of the supported sections for easy internal access at any joint.

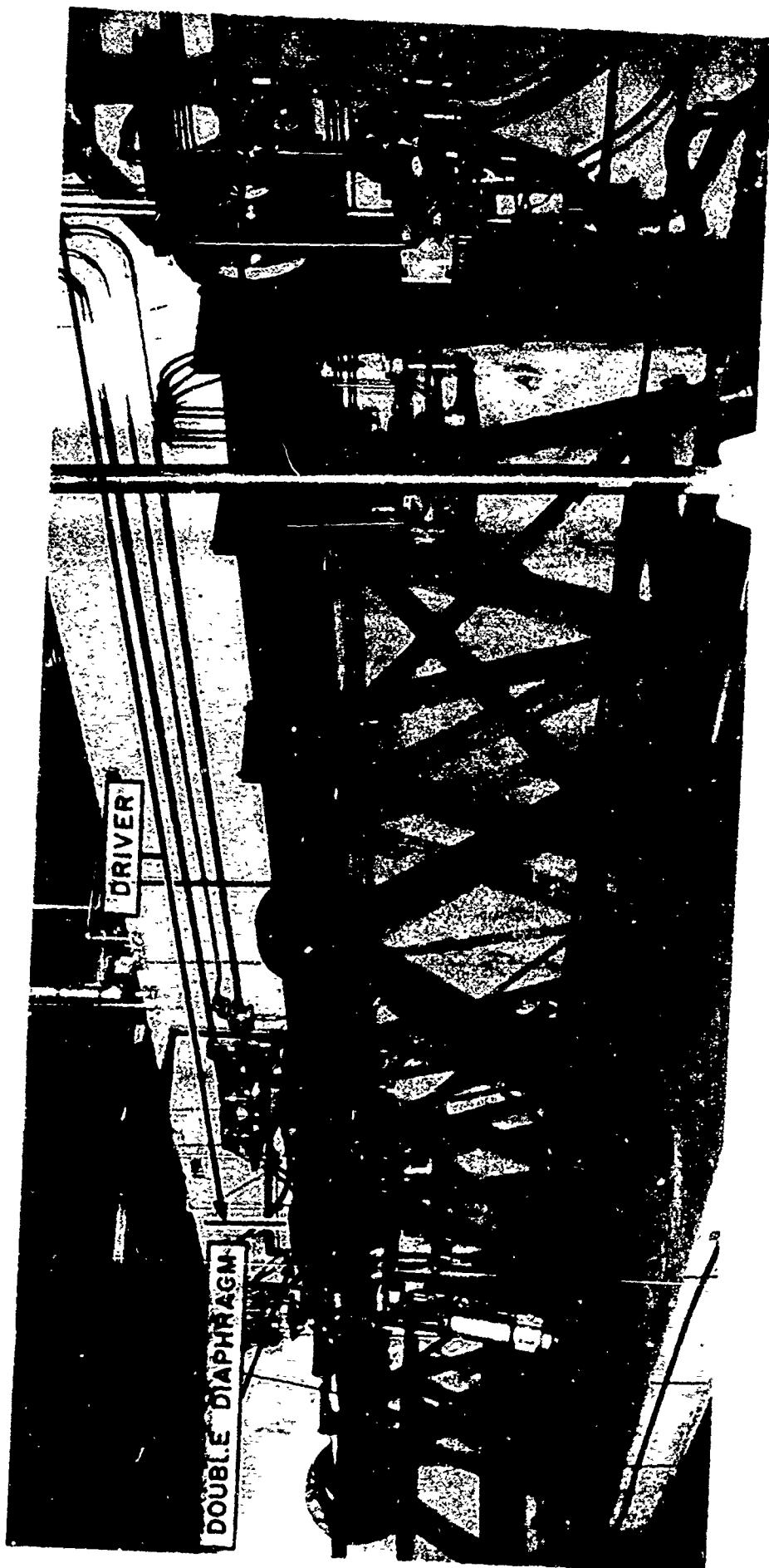


Figure 2. Driver and Double Diaphragm Sections

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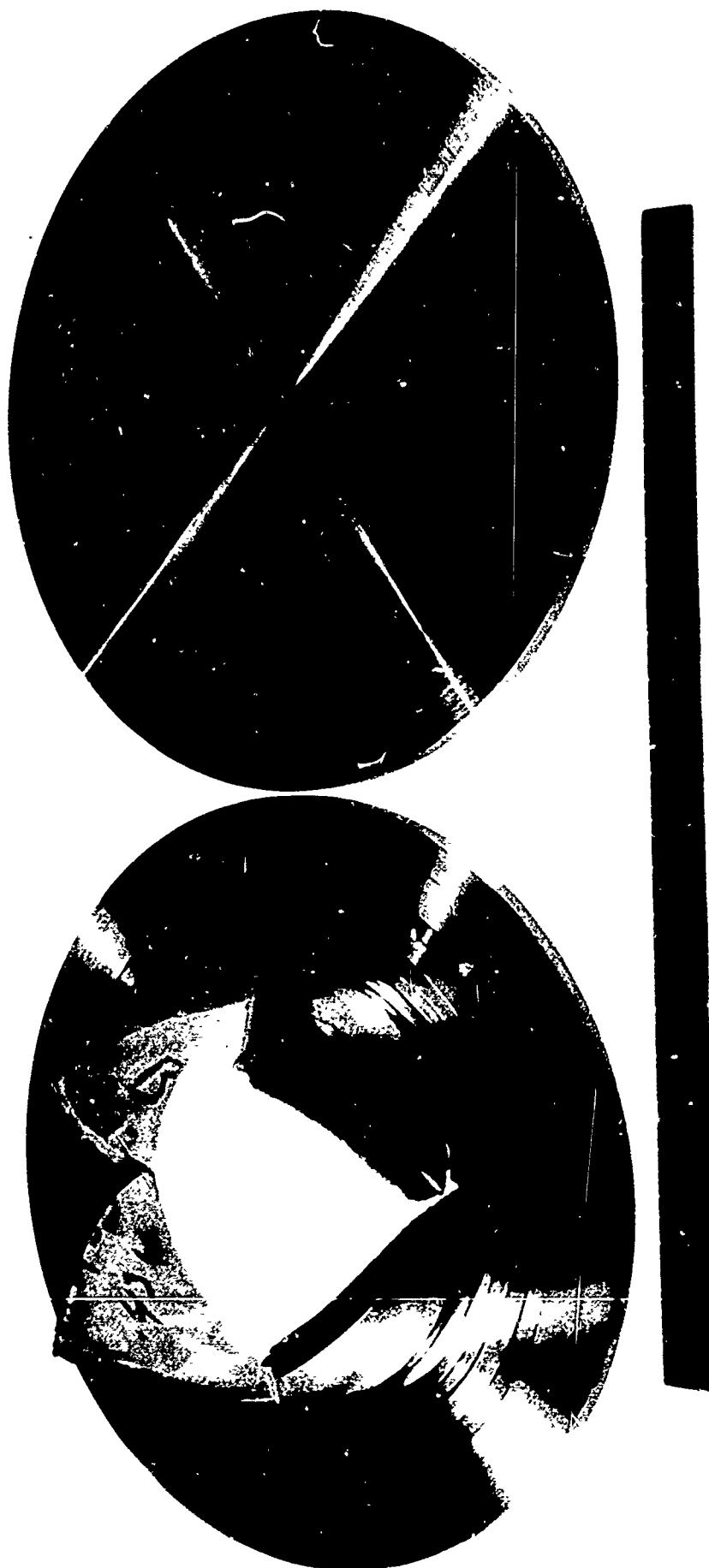


Figure 3. Rupture Diaphragms



Figure 4. Driven Section

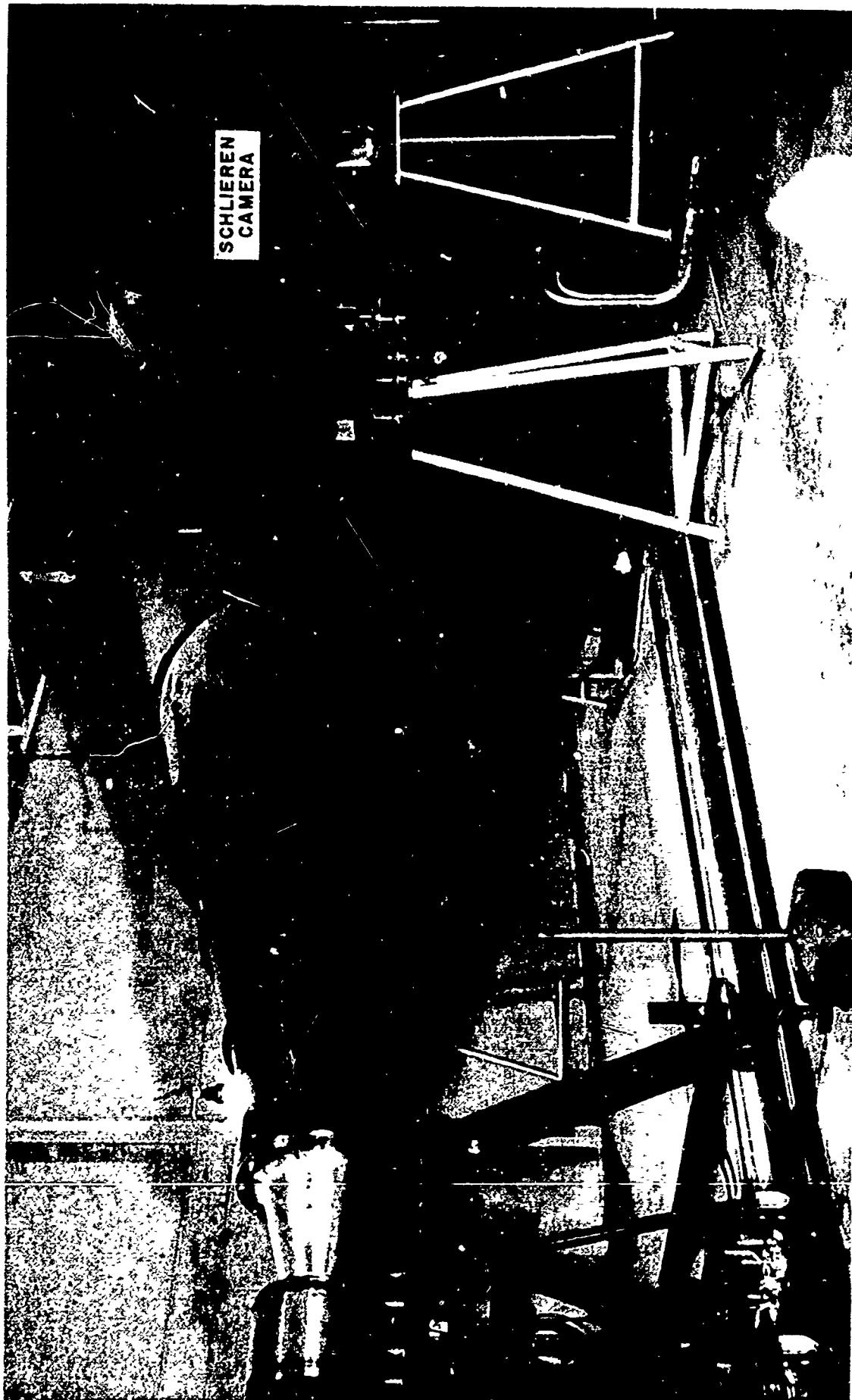


Figure 5. Nozzle Section and Expansion Tank

SECTION IV

DATA RECORDING AND INSTRUMENTATION CONTROL

1. CONTROL ROOM

The control room is isolated from the shock tunnel to provide maximum safety for all personnel operating the facility. This control room houses the central control console and the data recording console, as shown in Figure 6. The data recording console is a 50-channel data system and contains most of the equipment required to obtain, monitor, and record heat transfer and pressure data. In addition, a group of one megacycle electronic counters mounted in the data recording console provides a means of obtaining shock-wave velocity.

A graphic display panel, located directly above the central control console, pictorially traces the relationship of control equipment to shock tunnel operation. The controls on the central console are arranged on individual panels, generally in order of usage. Grouped together on the console are primary power, hydrogen pressurization, motor controls, evacuation, flush, charge driver, and charge driven section panels. In addition, a console-mounted tape recorder with microphone provides a means of monitoring control operations.

2. DATA RECORDING EQUIPMENT

Both pressure and heat transfer rate outputs from the model-mounted gages are fed through a switching panel directly to 25 Analab Instrument Corporation dual-trace oscilloscopes. Recorded data are photographed by scope-mounted Polaroid cameras, as shown in Figure 7.

A single pass Schlieren system is used to photograph flow patterns about a test model in the shock tunnel. With this mirror Schlieren system, either a continuous light source or an intermittent spark source can be used, and the corresponding photographs can be taken by a Wollensak Optical Company Fastax camera (8000 frames per second) or a simple box camera.

3. PRESSURE DATA

Pressure data is obtained from piezoelectric pressure transducers inserted in the test model or from probes. The pressure transducers, fabricated by Air Force Aero Propulsion Laboratory personnel, consist of a piezoelectric crystal that generates the signal output and which is housed in a case that is 0.375 inch in diameter and 0.5-inch long. This barium titanate crystal is mounted on a brass 0.001-inch shim stock diaphragm. The transducers are dynamically calibrated in an air shock tube. They have rise times of approximately 20 microseconds, and are capable of responding to pressures of the order of 0.001 psia. The accuracy of the measured pressure data is ± 5 percent. A typical calibration curve and output trace are shown in Figure 8. The high impedance output signal from each transducer is coupled through the switching panel, impedance matching devices, and signal amplifiers (if required) to oscilloscopes and Polaroid equipment. Electrometer cathode-follower circuits contained on each impedance matching device provide low impedance outputs suitable for oscilloscope readout. Signal amplifiers increase signal strength if necessary. A ground panel provides a means of remotely discharging capacitors within each impedance matching device, while battery panels provide power for the signal amplifiers.

4. HEAT TRANSFER RATE DATA

Heat transfer rate data is obtained by sensing the transient surface temperature of the model with a thin-film resistance thermometer gage. The heat transfer gages are fabricated by Air Force Aero Propulsion Laboratory personnel and consist of a 0.25-inch diameter Pyrex base on which a 0.20-micron thick layer of Hanovia Liquid Bright Platinum is brush-applied. This platinum strip has a resistance value of approximately 50 ohms. The gage is fired in a kiln to fuse the platinum and Pyrex, resulting in a thin sheet of metal backed up by an infinitely thick insulator. When a constant current of 50 millamps is applied to the gage, the output voltage (which is dependent upon the film's resistance) varies with temperature. This gage is used exclusively for the heat-transfer rates which are less than 880 BTU/ft² sec. These thin-film gages are dynamically calibrated by means of a pulse technique. For this calibration, the gage is connected to one leg of a bridge circuit, a known amount of electrical energy is pulsed through it, and the corresponding voltage-time response is recorded.

Heat transfer rate output from the gages is fed through the switching panel directly to the oscilloscopes and Polariod photographic equipment. The heat transfer panel, located in the central control room, maintains the balance of the current and voltage across each bank of heat gages.

The operation of the heat gage is based on the fact that the metallic film is thin enough to permit heat to diffuse through it very quickly (diffusion time of the order of 10^{-12} sec). For all practical purposes, then, the film senses the surface temperature of the insulator. The depth of heat penetration into the Pyrex base is small compared to the thickness of the base; therefore, the Pyrex base is treated as a semi-infinite heat sink. The heat flow into this base is treated as one-dimensional flow normal to the surface. Solution of the one-dimensional heat conduction equation yields

$$\dot{q}(t) = \frac{1}{2} \sqrt{\pi (k_p C_p)_{\text{glass}}} \left[\frac{T(t)}{\sqrt{t}} + \frac{1}{\pi \sqrt{t}} \int_0^t \frac{\sqrt{\lambda} T(t) - \sqrt{t} T(\lambda)}{(t - \lambda)^{3/2}} d\lambda \right]$$

and when written in terms of voltage output and made amenable to a trapezoidal type integration on a digital computer, the equation becomes

$$\dot{q}(t) = \frac{1}{k'' \pi E(f_0)} \left\{ \frac{2E(0)}{\sqrt{t}} + \frac{4}{\sqrt{h}} \sum_{m=0}^{m=n-1} [E(m+1) - E(m)] [\sqrt{n-m} - \sqrt{n-m-1}] \right\}$$

where $E(f_0)$ = initial voltage applied to the heat transfer gage

$E(0)$ = voltage from a chosen base line to base line of the scope trace

h = interval width

m = reading point

n = number of intervals

t = total time of trace

k = thermal conductivity

k'' = gage calibration factor

For a more comprehensive explanation of the heat transfer rate equation and the methods of application, see References 4 and 5.

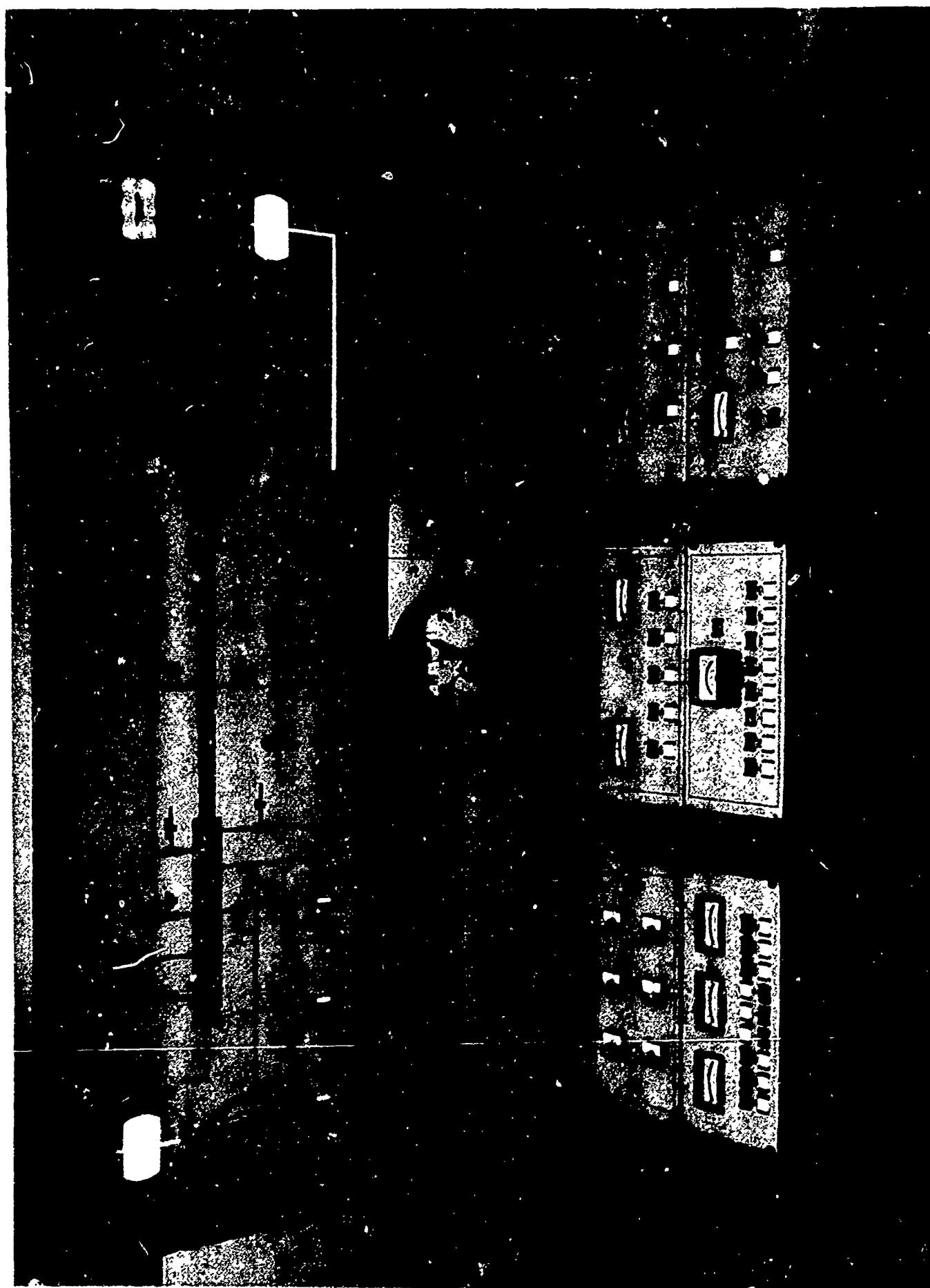


Figure 6. Control Panel



Figure 7. Data Acquisition Equipment

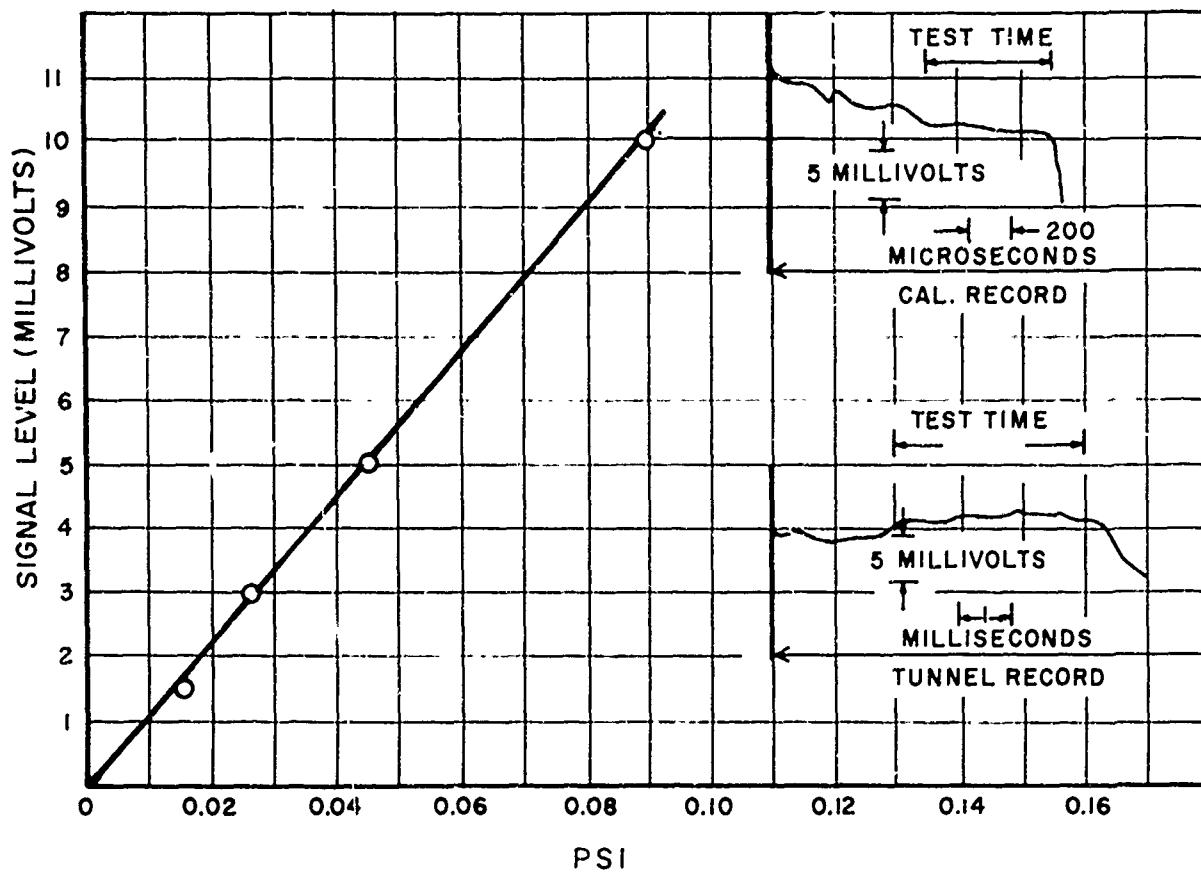


Figure 8. Pressure Transducer Calibration and Tunnel Record

SECTION V

ANTICIPATED PERFORMANCE

1. RESERVOIR CONDITIONS

Since the initial driven conditions (P_1 , T_1) are closely monitored during a test before a diaphragm ruptures, and since the shock Mach number (M_S) is easily measured during the test after the diaphragm bursts, the reservoir conditions can be calculated precisely. The driver is structurally limited to 30,000 psia but the 2:3 driver/driven diameter ratio reduces the effective driver pressure to 60 percent of the actual driver pressure (Reference 6). The SCRAMJET testing program requires maximum possible test time, which dictates "tailored operation." In turn, cold driver operation is then limited to the "tailored" shock Mach numbers of helium ($M_S = 3.7$) and hydrogen ($M_S = 6.0$). Reservoir conditions for these two shock Mach numbers are shown in Table I.

TABLE I
Tailored Reservoir Conditions (4-inch ID Driver)

RESERVOIR PARAMETER	HELIUM DRIVER	HYDROGEN DRIVER
Pressure (Atm.)	785	1125
Temperature ($^{\circ}$ R)	4059	7385
Density (lb/ ft^3)	7.64	5.96
Enthalpy (BTU/lb)	1108	2219
Entropy (BTU/lb- $^{\circ}$ R)	1.72	1.40

2. TEST SECTION CONDITIONS

A computer program has been developed by Air Force Aero Propulsion Laboratory personnel for the shock tunnel which performs a real gas calculation of the conditions in the reservoir, in the test section, and behind a normal shock in the test section, based on equations for thermodynamic properties from Reference 7. The calculations are made assuming chemical equilibrium and isentropic expansion of the test gas in the nozzle. By varying the reservoir pressure, by using different driver gases, and by using the three available nozzle throat inserts, we can obtain Mach numbers in the range 18 to 35. Also available are smaller exit nozzle sections which can be used to extend the lower Mach number limit to approximately $M_{\infty} = 8$. However, the anticipated performance figures presented in this paper are based solely on the 120-inch nozzle exit section.

The Reynolds number per foot is shown in Figure 9. As can be seen, the maximum Reynolds number per foot presently attainable is $Re/ft = 1.2 \times 10^6$ at a $P_4 = 30,000$ psia with helium driver (with a minimum h_5 and nozzle expansion).

Conversely, the minimum Reynolds number per foot is $Re/ft = 3.5 \times 10^4$ at a $P_4 = 1000$ psia, with hydrogen driver (with a maximum h_5 and nozzle expansion).

The tunnel's "tailored" performance with respect to a typical air-breathing hypersonic flight corridor is shown in Figure 10.

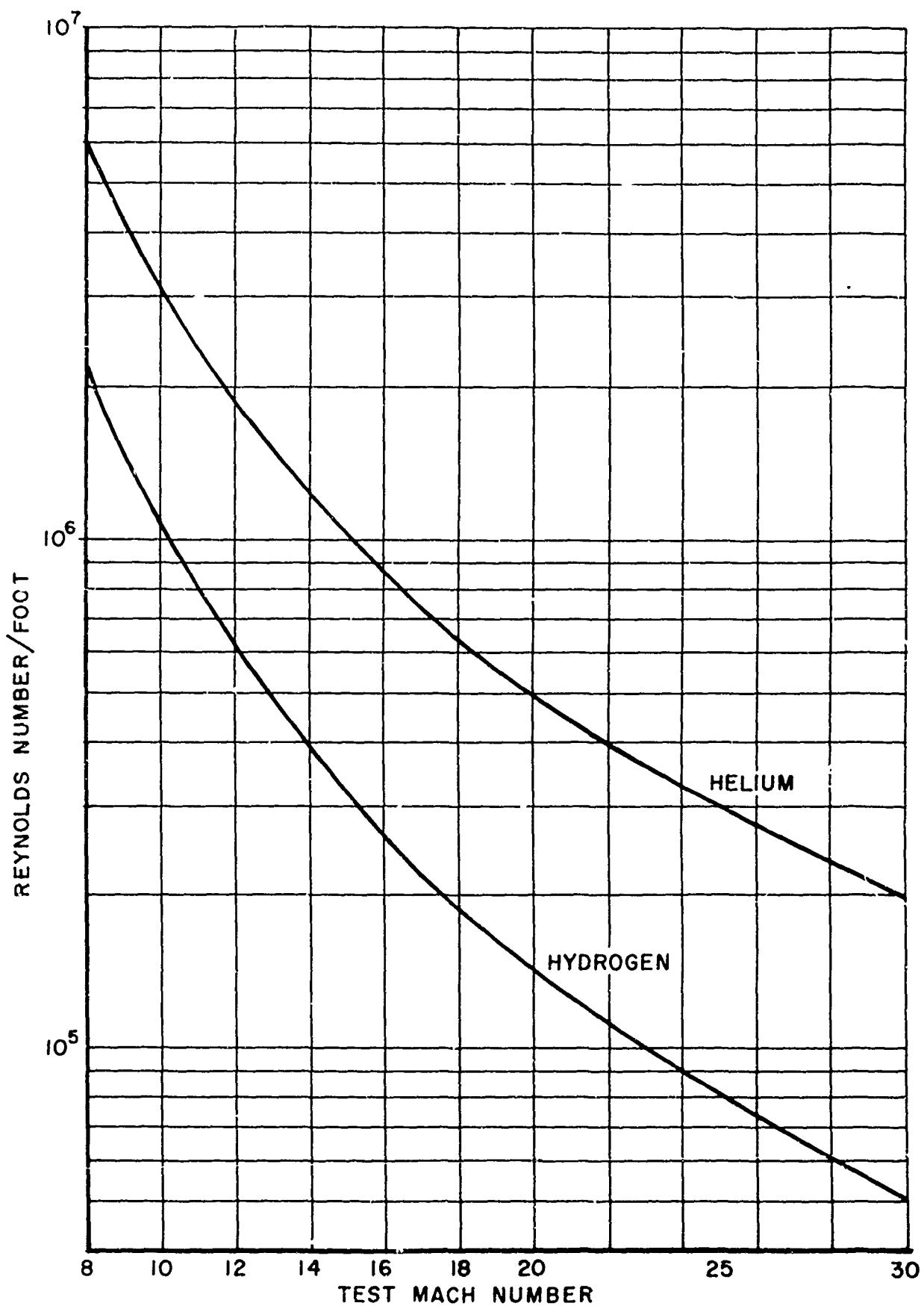


Figure 9. Reynolds Number Per Foot vs. Test Mach Number,
"Tailored" Operation and 4-Inch Driver

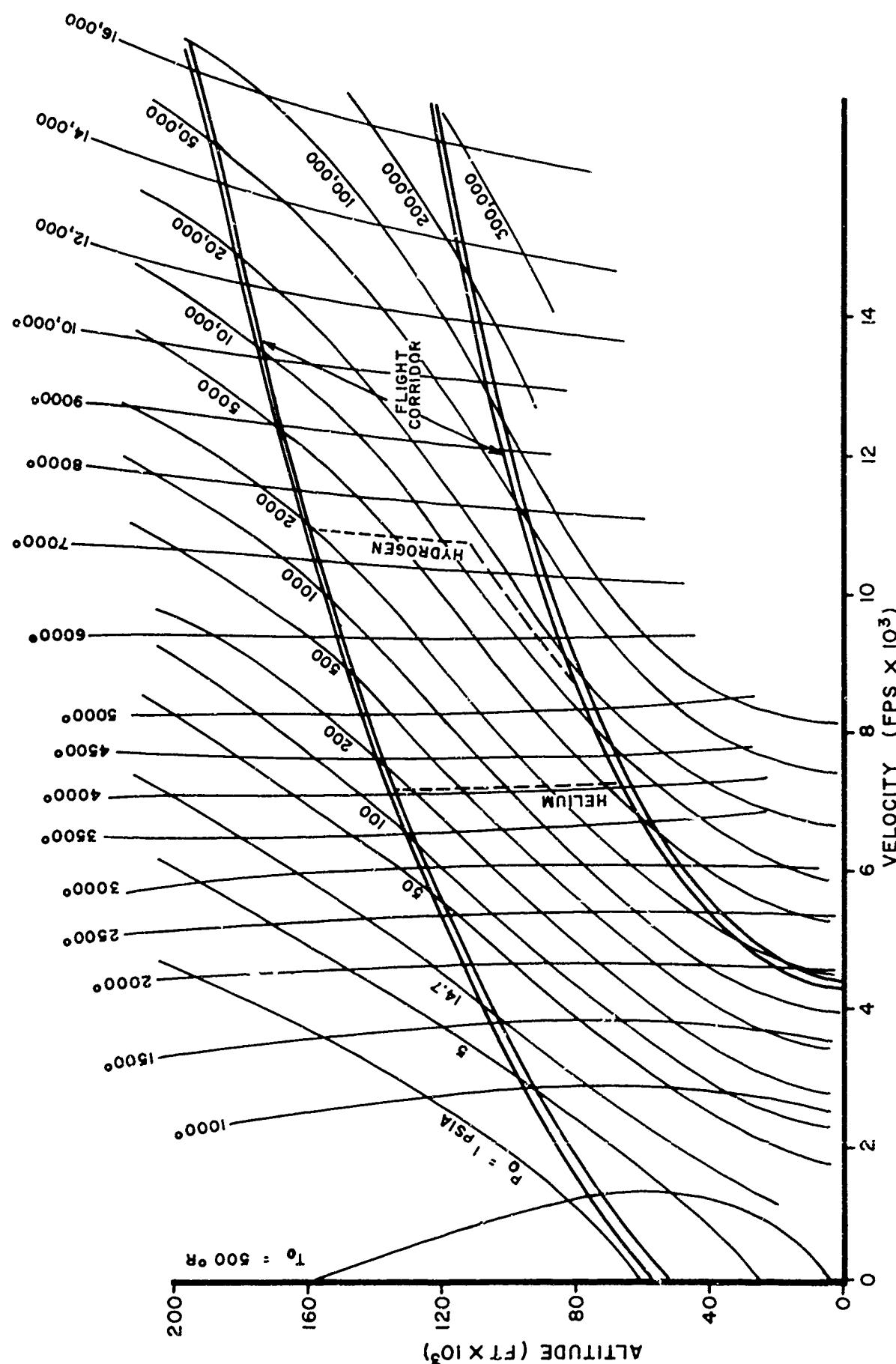


Figure 10. "Tailored" Performance in Typical Airbreathing Corridor

SECTION VI CONCLUSIONS

To date, calibration and shakedown of the tunnel and the associated support equipment have consisted of cold helium tailored runs for driver pressures ranging up to 27,000 psia. A total pressure rake spanning the 10-foot test section has been used on all runs to obtain an accurate picture of the Mach number variation across the test core. High-pressure runs on an instrumented 10 degree cone have also been accomplished. Results so far indicate that the 120-inch shock tunnel will have an excellent Mach number - Reynolds number simulation capability for the Mach number range of 18 to 35.

Future modifications to the facility will extend the range to lower Mach numbers (approximately Mach 8 to 10) and refinement and miniaturization of existing instrumentation will increase the facility's capability of testing the most advanced SCRAMJET engine components and large-scale integrated propulsion systems.

LIST OF REFERENCES

1. C. E. Witliff, M. E. Wilson, and A. Hertzberg. "The Tailored Interface Hypersonic Shock Tunnel." J. Aero/Space Science 26, No. 4. April 1959.
2. R. F. Flagg. Detailed Analysis of Shock Tube Tailored Conditions. AVCO Technical Memorandum RAD-TM-63-64. Research and Advanced Development Division, AVCO Corporation, Wilmington, Massachusetts. September 1963.
3. I. I. Glass, and J. G. Hall. Handbook of Supersonic Aerodynamics - Section 18-Shock Tubes. NAVORD Report 1488, Vol. 6, December 1959.
4. R. J. Vidal. Model Instrumentation Techniques for Heat Transfer and Force Measurements in a Hypersonic Shock Tunnel. WADC TN-56-315 Cornell Aeronautical Laboratory Report No. AD-917-A-1. February 1956. AD-97238.
5. D. K. Heron. A Method for the Reduction of Heat Transfer Data from Thin Film Heat Transfer Gauges Used in Shock Driven Facilities. AVCO Corporation T.R.R. - R520 - 63-6. AVCO Research and Advanced Development, Wilmington, Massachusetts. September 1963.
6. S. C. Lin, and W. I. Tyfe. Low Density Shock Tube for Chemical Kinetic Studies. AVCO Report 91, AVCO Research Laboratory, Everett, Massachusetts. July 1957.
7. W. E. Moeckel, and K. C. Weston. Composition and Thermodynamic Properties of Air in Chemical Equilibrium. Lewis Flight Propulsion Laboratory, Cleveland, Ohio. April 1958.
8. J. A. Copper. The Hypervelocity Impulse Tunnel: Facility Description and Expected Performance. Douglas Aircraft Company Report SM 4 1377. Missile and Space Systems Division, Douglas Aircraft Company, Inc., Santa Monica, California. November 1962.

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